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Recognition and Correction of Sonic Fatigue Damage in Fighter Aircraft

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Abstract

High performance aircraft that fly aggressive low altitude or high-angle of attack maneuvers will experience high acoustic loading, especially in the presence of external stores. These high acoustic loads can lead to rapid sonic fatigue in thin gage metallic structure. As a means of prolonging fatigue life, while at the same time restoring structural load carrying capability in damaged structure, composite bonded repairs which incorporate damping layers have been developed. These damped composite bonded repairs are installed on F-15 flight test aircraft. This paper describes ways to recognize sonic fatigue damage, and the design and flight test results of representative damped patches.

Introduction

Military services spend millions of dollars annually to repair cracks in fighter aircraft. These fatigue failures result in significant system down time and reduced operational readiness. Fatigue problems can be divided into two categories: primary structure and secondary structure.

Primary structure cracks are typically the result of maneuver and other static loading which require bolted metal doublers to reinforce the damaged structure. These repairs can be complex, costly, and heavy. When a bolted repair is not possible, replacement of the cracked component can bring additional complications and cost.

An alternative life enhancement technique for primary structure is the bonded composite patch. These patches offer a more efficient way of repairing cracked structure by avoiding drilling new holes and offer weight and cost advantages over bolted repairs. Composite bonded repair technologies have been proven effective on transport aircraft [1-4]. However, this technology is just starting to be applied to fighter aircraft [5-11], where dynamic environments play a more prominent role in the cause of fatigue damage.

Secondary structure failures are often the result of sonic fatigue, which is defined as cracking due to dynamic response from fluctuating acoustic pressures. Structures, which are susceptible to sonic fatigue include, large, flat or thin panels, panels with stress concentration factors (Kt) and structural surfaces exposed to high noise, high temperature, and high mean static stress. These loads are ever increasing as a result of:

- More agile aircraft that can fly at high angles of attack and use vortex impingement to maintain controlled flight.
- More powerful and responsive engines that exhibit higher inlet acoustic levels, exhaust velocities, and exhaust temperatures, plus the ability to perform rapid throttle changes.
- Carriage of fuselage mounted stores and pods which generate turbulent wakes and shocks during high dynamic pressure (Q) flight.
- Blast pressures from large caliber, rapid fire cannon.
- Oscillating, turbulent flows in weapons bays and their wakes.

Paper presented at the RTO AVT Specialists' Meeting on "Life Management Techniques for Ageing Air Vehicles", held in Manchester, United Kingdom, 8-11 October 2001, and published in RTO-MP-079(II).

On fighter aircraft, secondary structures are lightly loaded during maneuvers, but they still experience cracking primarily due to dynamic loads from buffet, flow-induced noise, shock/boundary layer interaction, and turbulence. Acoustic loads are highest during high speed and low altitude missions for the lower surface and during high angle of attack maneuvers for the upper surface. On the lower surface, turbulent flow forms around external stores, Figure 1, and will be used as an example source of damage is discussing repairs. This noise is characteristically high in frequency (200 to 1000 Hz) and broad band in nature as will be shown in Figure 4.

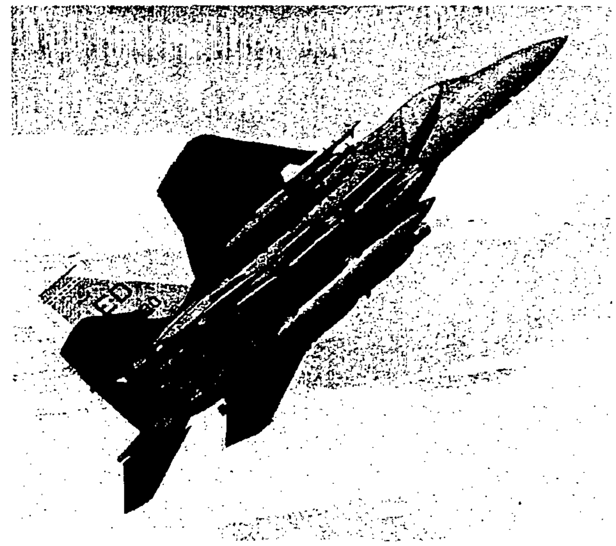

Figure 1. On the Lower Surface, Turbulent Flow Forms Around External Stores

For these types of damage, the addition of damping to the structure is known to be an effective solution. Furthermore, combining adhesive bonded composite patches with damping treatments provides robust and effective repair designs for both damaged, and as yet undamaged structure.

Sonic Fatigue Damage Recognition

Sonic fatigue damage may be recognized by the type of structure suffering damage and by the location of the damage. The majority of structure that cracks due to the acoustic loads are thin gage metallic structure; such as, fuselage skins, access doors, non-load carrying fairings, leading edges, and secondary support structure; such as, stiffeners, stringers, shear clips, and brackets. In thin gage metallic structure, cracking tends to occur at stress discontinuities arising from chem-mill radii or at fastener holes. The out-of-plane bending response of a panel causes the highest stresses and bending moments to occur at or near the edges of the panels where such features and support structure are located. The critical locations are those which experience the flow effects from the sources enumerated above. They include light panels in engine inlets, behind engine exhausts and inlets, near gun muzzles, within weapons bay cavities, or in the wakes of turbulence producing features.

For maneuver load (or static) fatigue considerations, increasing part thickness to reduce stresses can be a solution. However, for dynamic (or sonic) fatigue problems this approach is generally ineffective because reduced stress is usually accompanied by increased stiffness, which tends to move fatigue problems into adjacent structure. For sonic fatigue, the addition of damping to the structure is known to be an effective fix. Furthermore, coupling adhesive bonded composite patches with damping treatments provide robust and effective repair designs.

Three examples are offered to illustrate these concepts.

(1) Chem milled inlet skin panels of 0.05 in. thickness began cracking after 60 hr of service. The skin was increased to 0.071 without chem.-mill, and the cracks subsided until a more powerful engine with different fan characteristics was installed. Then, stringer flanges began cracking within 250 hr of the change. The stringers were changed to titanium, and within 200 additional flight hours, the rivets attaching the skins to the stringers began failing. This is a typical example of chasing the damage as a result of selective structural strengthening.

(2) A small fuselage skin panel, Figure 2, incident to the engine exhaust plume exhibited cracks after 150 hr. The cracks were stop drilled and small riveted doublers were applied. Within another 200 hr the cracks were appearing from beneath the doublers. Other aircraft were repaired with larger riveted doublers containing a myriad of "field rivets." In a short time cracks began to occur in the doublers at the rivets. This example typifies the inability of mechanically fastened doublers to arrest sonic fatigue cracking.

(3) After installation of a blunt ended pod beneath the fuselage, aircraft began suffering skin cracking and door damage after high Q sorties. After one particularly demanding occurrence, an aircraft returned with the damage shown in Figure 3. This is an example of the damage that can be caused by turbulent wakes and shocks.

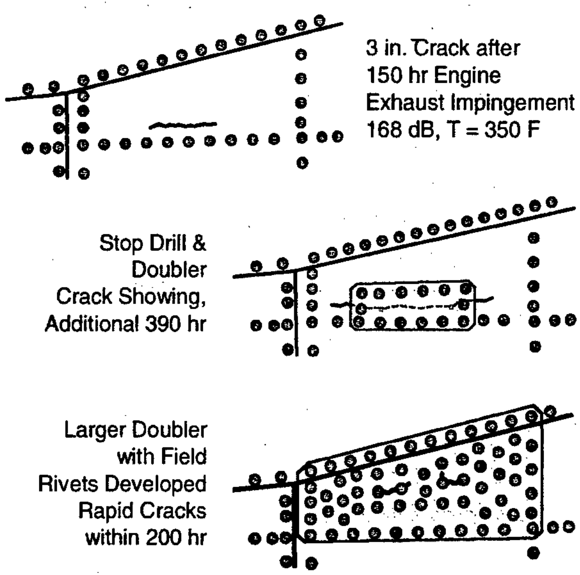

Figure 2. Failed Traditional Sonic Fatigue Repairs

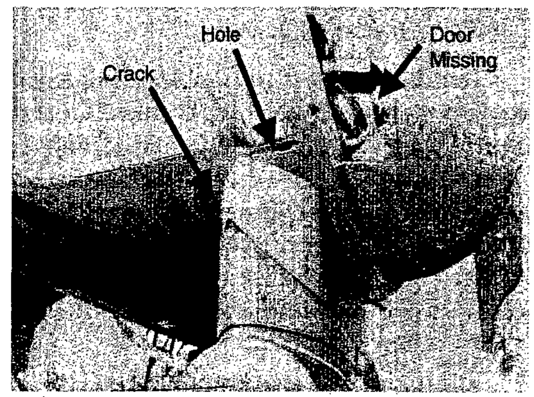

Figure 3. Sonic Damage After Prolonged High Q Flight

Typical Acoustic Loads

The increase in the acoustic spectrum on the fuselage skin at the critical locations from the addition of the pod may be seen in Figure 4. It may be noted that the spectrum levels are high over a wide frequency range. Therefore, strengthening the panel, which will increase its natural frequency, will not help, and in fact a change from 300 Hz to 500 Hz would be detrimental.

Other excitation sources mentioned herein have levels of similar magnitude. Engine exhaust plume spectra may be in the 165 dB to 170 dB range, and relatively flat over a wide range of frequencies. Acoustic spectra from engine inlets and weapons bays will

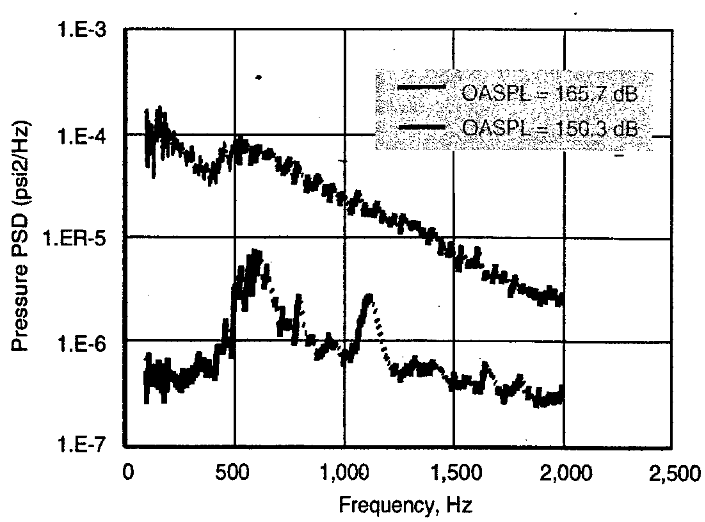

Figure 4. Incident Sonic Spectrum, With and Without Stores

have high broad-band levels, and in addition contain pure tones of very high amplitude. However, it will be impossible to tune the structural frequencies to avoid the tones because they change frequency with engine speed in the inlet case, and with Q in the weapons bay case.

Sonic Fatigue Repair Types

There are no special repair guidelines or maintenance procedures for sonic fatigue damage. A traditional repair using a fastened doubler is still the repair of choice at all levels of maintenance. The repair procedure involves stop drilling the end of cracks or the complete removal of the damaged area. The doubler is then matched drilled to the underlying structure. A sealant is spread over the doubler and the repair is bolted in-place on the aircraft. Doublers are effective in increasing the stiffness of the subject panel. This increased stiffness is often sufficient to prevent the repaired panel from further cracking if the panel were loaded in plane.

Reinforcement - The problem with repairs that involve stiffening only is that fatigue damage often appears on adjacent structure, because the dynamic energy is still present. Doublers mass load the sub-structure and cause higher dynamic bending moments at the panel edges, which may change the response of near-by panels. See Figure 5, which shows the reduction ratio in stress for the patch application as a function of doubler thickness. It may be noted that as the doubler thickness increases, the reduction ratio at the panel edge decreases below 1.0, which indicates an actual stress increase. For example, if a panel is repaired with a doubler of equal thickness, the stress at the panel center is reduced, but the stress ratio at the edge is 0.75, which is a stress increase of 33%. This will likely cause the failure to migrate to the substructure.

Another reinforcement approach involves replacing the failing panel with a thicker one, or a panel of a stiffer, more fatigue resistant material. The result will likely be the same with failures migrating to the substructure as has been noted in one of the example cases in a previous paragraph.

Damping - Damping provides a means for reducing the amplitudes of vibration through dissipation of energy. In built up metallic structure, the damping present is attributable to the minute slippage between fastened assemblies and through internal hysteric material damping. One method to enhance damping in structures is to introduce viscoelastic materials to the assembly. The most efficient form of viscoelastic damping for thin sheet metal structures is constrained layer damping[12].

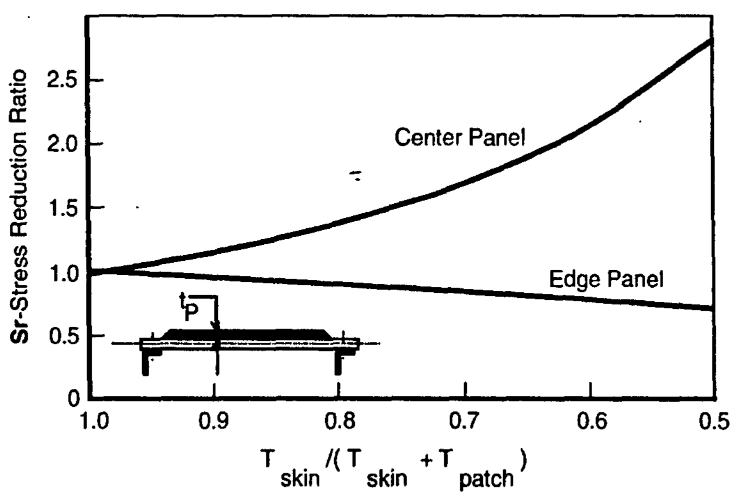

Figure 5. The Effect of Skin Reinforcement on RMS Panel Response

In linear dynamics systems, damping is inversely proportional to the response. Hence, as the damping increases the peak response decreases proportionally. Under broad band random vibration, the response is inversely proportional to the square root of the damping. Hence, the damping would have to be quadrupled in order to decrease the root mean square (RMS) response by 50%. This relationship is shown graphical in Figure 6. Referring to the figure, if the damping is increased by a factor of 6 to 7, a factor which can be easily achieved, the stress reduction in the panel is 2.5 as with the doubler. Unlike the simple doubler, the stress at the edge is also decreased by a similar amount.

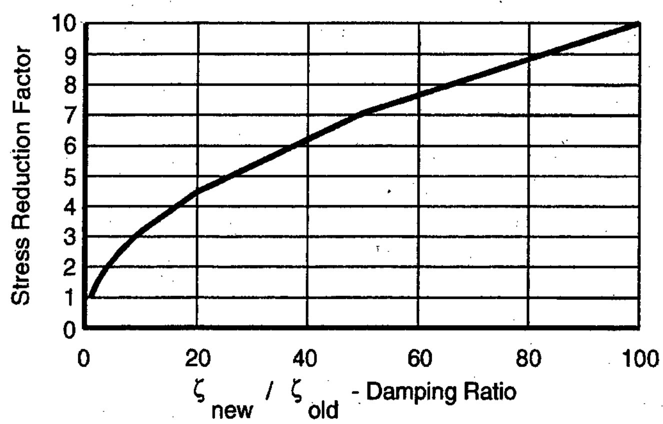

Figure 6. The Effect of Damping on Single Mode RMS Response

Bonded Repairs

The discussion has shown that damping is effective in lowering loads in and around the panel, where a doubler (or patch) alone reduces stresses in the panel center, but can increase load going into the substructure. These conclusions led to the investigation of composite bonded patches that incorporate damping. Damping reduces the adverse effects of doublers on substructure while continuing to restore load-carrying capability. A complete test program was developed for investigating the effectiveness of composite bonded patches as repairs for sonic fatigue damage. In reference (5), the first phases of this test program were described. These were tests for the development of damped repair technology, including coupon beams, sub-element vibration, and acoustic panel tests. The next phase of development involved the application of patches to high response skins and flight testing on fighter aircraft.

The process of designing bonded repairs must take into account the durability of the damaged structure, the repair, and the adhesives used to attach the repair to the structure. Millions of cycles can occur in a few flights for acoustic loadings and current state-of-the-art repairs for maneuver load damaged structures typically do not account for sonic durability. They are usually based only on restoring static strength. Maneuver load repair approaches can be inadequate for acoustic loaded structures where static loads are often much less significant and cracking typically occurs in a short period of time, often before the airframe has experienced 500 flight hours.

Installing damped bonded repairs to improve damping characteristics and to restore structural stiffness introduces challenges. Issues associated with surface preparation, bondline integrity, and corrosion protection must be addressed. Access and sealing must be considered to allow using vacuum bag and heat curing. Post repair NDE must be able to verify integrity as well as monitor repair performance throughout the life cycle.

Another consideration is cost. Although, detailed cost studies have not been performed, it has been suggested that the replacement of sheet metal parts is more cost effective than installation of bonded repairs. However, a replaced part will still have the fragility and will be prone to fatigue. Hence, standard composite bonded repair methods need to be altered to make them more attractive for application to thin gage sheet metal panels.

Some recent investigations into damped repairs can be found in Ref. (5-7). The use of damping in composite bonded repairs has been given more attention in recent years. As part of the Durability Patch program [7], the Air Force Research Lab has contracted with Boeing and CSA Engineering to develop and test damped composite patches suitable for high-cycle fatigue life extension. Also, as part of this program they have developed a low cost flight instrumentation module for diagnosing fatigue problems. The device, called a Damage Dosimeter, has a programmable interface and can be used to measure strains under dynamic conditions for long term monitoring. Add-on damping and composite bonded repairs have also been investigated at Aeronautical & Maritime Research Laboratory (AMRL) in Australia [6] for use in the outer skin of the engine inlet on RAAF F/A-18 aircraft. Their work has shown the feasibility of using a damped repair concept.

A just completed test program at AFRL, and supported by Boeing and AMRL, has substantiated mush of the discussion of this paper 7. Cracked panels were patched using various concepts, and the acoustic exposure continued to monitor crack growth. The results showed the addition of damping to be the best method of repair. Constrained layer damping applied to about 80% of the panel area performed the best, but back-side access is required for installation, and a patch is required for the crack in addition. Of the front access-only repair concepts tested, patches of the type described in the remainder of this paper performed best.

The demonstrated technology as described in this paper is transition ready for fleet wide use. Development has progressed through tests and studies to make the incorporation of damping suitable for bonded repair applications.

Structural Demonstrations

As a demonstration of the damped patch technology, two structural applications were chosen that had a history of sonic fatigue due to their proximity to external stores. The first was a center fuselage chine door, and the second was a lower surface skin panel. The latter panel was instrumented with sensors to monitor the structural response. The flight tests will be discussed below.

Center Fuselage Chine Door

On June 17, 1998, an F-15 flew for the first time with a damped composite bonded patch installed on the center fuselage chime door. The patch is a demonstration of pre-emptive type repair technology suitable for sonic fatigue prone structures. This door, and other similar ones, have had a history of damage including skin cracks, latch wear, and hinge fitting cracks. Figure 7 shows the door and patch prior to bonding and as installed on the aircraft. The patch consists of Textron Boron/Epoxy 5505 with a 3M adhesive integrated into the laminate. The patch was then bonded with paste adhesive onto the surface prepped door and cured in an autoclave for a full bonding. This door has been operational on a Boeing/USAF flight test aircraft for over two years.

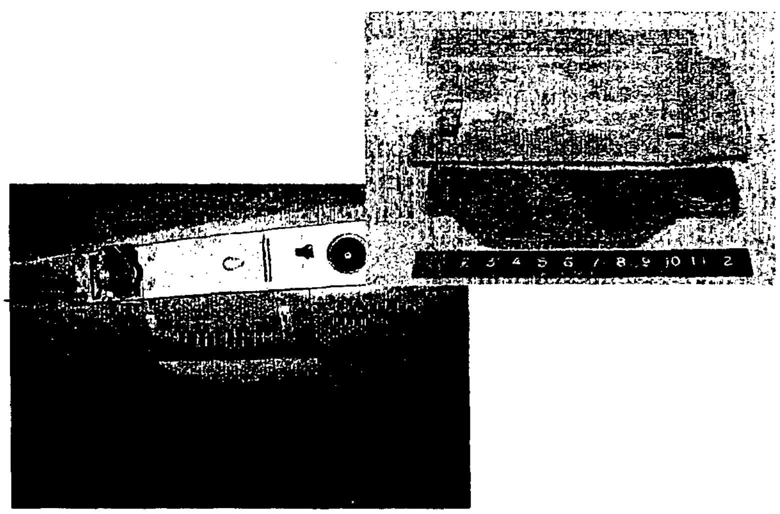

Figure 7. Center Fuselage Chine Door w/ Damped Patch

Lower Surface Skin Panel

The second demonstration is a lower surface skin panel on the F-15, Figure 8. This panel is part of the lower nacelle skin assembly and experiences high dynamic response during low altitude and high-speed maneuvers. The panel is made of 2024 Al. It is 0.071 inches thick with chem-milled pockets to 0.060 inches and 0.044 inches, and has experienced cracks in the former and in the panel chem-mill boundary. According to Figure 5, this panel is a good example of how a doubler could aggravate the problem if not properly designed to lower loads at the panel edge.

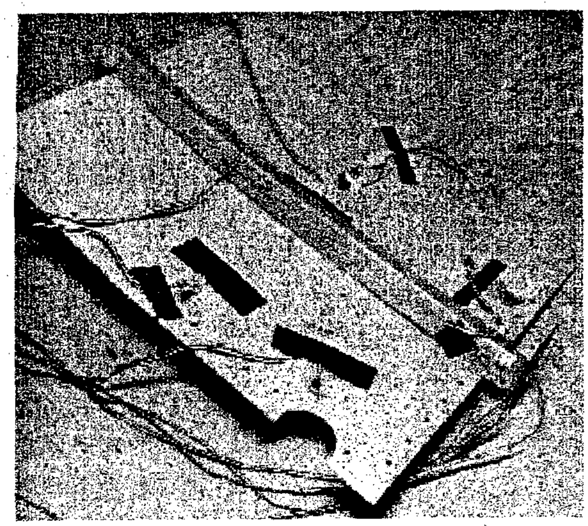

Figure 8. Lower Nacelle Skin Panel Removed from Aircraft Showing the 487.3 Former

Thc purpose of the damped doubler is to reduce loads in the skin and in the former through damping. The design was verified through finite element analysis. The damping performance of the patch was determined through modal strain energy analysis[14]. The panel was analyzed using random response analysis to determine overall RMS loads reduction, Figure 9. The contour plot shows how the stresses in the skin panel are expected to change. Large reductions are expected in the skin panel with smaller reductions at the edges. The modal damping predicted by the analysis was on the average of $\zeta = 5\%$ for the lowest 5 modes at the temperature of interest, which is an increase of over 4.

A composite patch system was designed with integral damping treatment, as a simple and low cost retrofit, two 10 by 5-inch rectangular patches of boron/epoxy with integral damping. The complete laminate contains Textron Rigidite Boron 5505, and a damping pack comprised of damping and constraining layers (patent pending). The final patches are shown as installed on the aircraft, Figure 10.

Normalized Principle Stress Reductions:

Location	Ratio (BL w/Patch to BL)
Lug Skin	0.45
Chem-Mill	0.4
Shear Clip	0.6

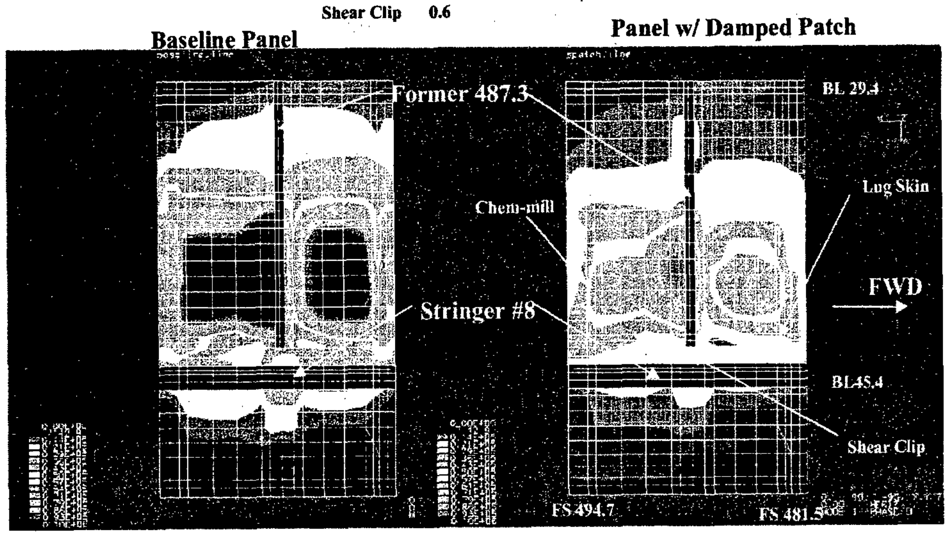

Figure 9. Random Vibration Analysis of Panel with and without Patch

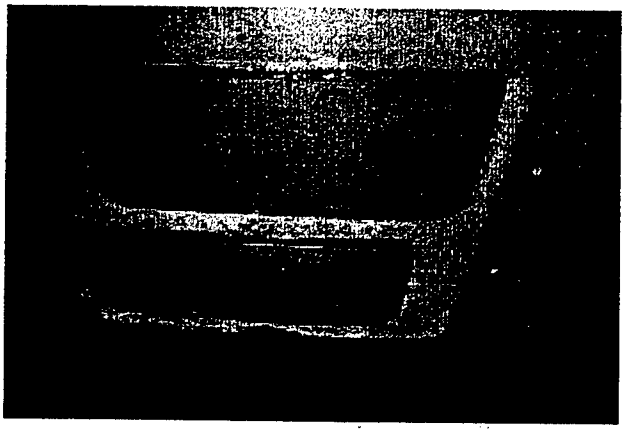

Figure 10. Patches Installed on A/C Prior to Painting – Looking Aft

Flight Test Program and Results - For the flight test, instrumentation was installed to interpret the change in the dynamic response of the panel. This instrumentation includes strain gages and accelerometers for measuring response, pressure and temperature transducers for measuring the environment. Flight-tests and modal surveys were conducted before and after patching. A similar flight plan was used for both the before and after tests, and similar external store configuration was used.

The initial flight tests established the baseline response and environmental factors such as temperature, acoustic spectrum levels, and quasi-static loading conditions. For instance, the temperature varied 20-F to 105-F. The absolute static pressure varied from 5-psia to 18-psia. The comparison of patched to unpatched response was made with the strain gages and accelerometers. On an overall basis, the acceleration showed the most dramatic response reductions for the patch configuration. These PSDs plots are shown in Figure 11.

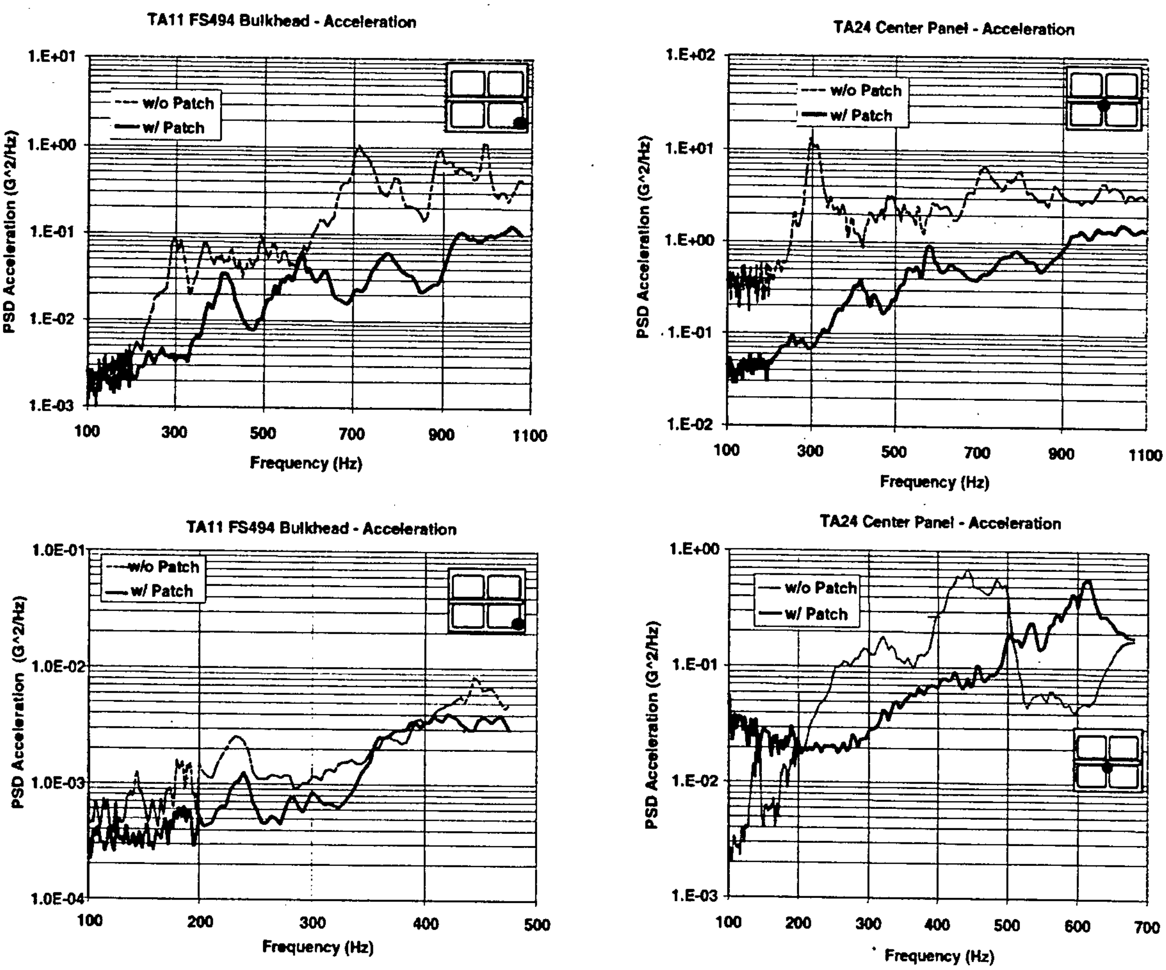

Figure 11. Comparison of Acceleration PSDs w/ and w/o Patch
(Note: The top figures are w/ external stores and the bottom figures are w/o external stores.)

For the clean aircraft (w/o stores) configuration, the center-panel response drops dramatically with the damped patch. The loads going into the bulkhead to which the panel attaches at FS 494.7 are noticeably reduced at the frequencies corresponding to the panel modes. Similarly, for the w/ stores configuration, the center-panel response drops dramatically with the damped patch. The loads going into the bulkhead to which the panel attaches at FS 494.7 are noticeably reduced at the frequency corresponding to the panel modes. On a closer examination, the RMS values for different frequency bands is listed in Figure 12. Note that better response reduction occurs in the 200 to 1000 Hz band than in the overall.

Measurand	Frequency Range (Hz)	RMS Acceleration G_{rms}		Percentage Difference
		W/o Patch	W/ Patch	
TA24	0-200	1.2	3.6	-198.39%
w/o stores	200-1000	4.5	3.2	28.94%
	1000-2000	15.1	23.0	-52.35%
	overall	17.6	24.1	-36.81%
TA24	0-200	8.8	3.4	61.41%
w/ Stores	200-1000	16.2	5.1	68.75%
	1000-2000	71.5	40.9	42.85%
	overall	78.9	42.2	46.52%
TA11	0-200	0.7	0.8	-1.97%
w/ Stores	200-1000	2.2	1.2	42.04%
	1000-2000	25.6	10.4	59.35%
	overall	26.0	10.7	58.70%

Figure 12. Comparison of RMS Acceleration Loads w/ and w/o Patch

Fleet Management Considerations

The types of damage described herein usually manifest themselves rather soon after the configuration begins full envelope flying, or after a change is mission, payload, or engine occurs. Further, it is typically not an isolated case, but will show up on most of the fleet within several hundred additional flying hours. For this reason it is prudent to patch the entire fleet as soon after a trend is noted as is practical. With this approach, the damping may be applied to the critical area, thus avoiding future damage with the addition of minimum weight and complexity. A patch will require more plies to restore strength after cracking initiates. This type of "pre-emptive" repair is recommended.

Design and application of a patch uses the same methods as other composite patching of metal structure, which may be found in the current literature. For the applications reported herein, the fiber portion of the patches was boron. However, carbon or glass fiber could be used with proper care. Surface preparation involves removing all existing paint, grit blasting, and application of silane and a primer. It is recommended that the patch be pre-cured and then bonded in place in a separate process using an adhesive that cures with only a modest temperature elevation.

The choice of the specific damping material to be used in the application depends on the temperature range to which it will be exposed. Actually, two temperature ranges should be considered; (1) a wider range within which the patch must survive without suffering brittle fracture or creeping, and (2) a narrower range within which the damping material must perform. The wider range for fighter aircraft is between –40C and 95C, whereas the performance range is between –20C and 50C. Vendor data is available for a substantial number of materials to aid in selection.

Summary and Conclusions

Rapid fatigue damage can occur in light weight structure that is subjected to high sonic loads. This type of cracking is identified by considering the type of structure suffering damage, and its location in an area of high sonic loading. It is important to identify the damage as sonic fatigue because the type of repairs typically employed for fatigue failures are not effective for sonic fatigue. The use of fastened metallic doublers or part replacement of thicker gage will likely chase the damage to adjacent locations. Replacing the parts with those of a more robust material will have the same result unless the adjacent structure and fasteners are of sufficient strength and durability.

Damped composite bonded patches have been developed for sonic fatigue applications that are low cost and easy to install. These patches are ideal for repair of cracked structure; as well as, being useful as preventative a retrofit against fatigue. General considerations have been discussed for the importance and effect of adding stiffening and damping to solve sonic fatigue. The flight test program discussed herein, has validated the applicability and performance of the damped patch concept as a long term solution for sonic fatigue damage.

Acknowledgments

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